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An autonomous system to detect both absolute and differential spacecraft charging aboard high altitude satellites, and to reduce those potentials before hazardous arcing levels are reached, is now being developed under the Air Force program on Space Systems Environmental Interactions Technology. Operation of the system is based on the empirical results of the Space Test Program SCATHA (P78-2) and NASA ATS-6 satellites, both of which successfully demonstrated the principle of safely reducing spacecraft charging levels by the emission of a low energy neutral plasma--effectively shorting the spacecraft and charged dielectric surfaces to the ambient space plasma. The Charge Control System, now being designed and built at Hughes Research Laboratories in Malibu, CA., will utilize a xenon-based plasma source capable of igniting within one second, and capable of emitting a quasi-neutral plasma containing more than 1 mA of ions. The spacecraft charging level will be detected by sensors similar to those that operated aboard SCATHA. Satellite frame potential (relative to the ambient space plasma) will be determined by an electrostatic analyzer capable of detecting both ions and electrons in the energy range 50eV-20 keV. Differential charging (relative to spacecraft frame ground) will be determined by a surface potential monitor utilizing two different dielectric surfaces, and covering a range of ± 20 kV with a response time of one second. A transient pulse monitor will detect the presence of exterior spacecraft arcing by measuring its near-field electromagnetic radiation. Automatic operation of the system will be accomplished by a microprocessor controller which will interpret the sensor data and activate the plasma source when predetermined threshold levels are exceeded. With a gas supply for more than 2000 hours of operation in space, the system may be expected to provide on-orbit spacecraft protection for up to 10 years. The system will be completed by the end of 1987, and is expected to be flight-tested at geosynchronous orbit in the 1988-1992 time period.

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A Charge Control System for Spacecraft Protection

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An autonomous system to detect both absolute and differential spacecraft charging aboard high altitude satellites, and to reduce those potentials before hazardous arcing levels are reached, is now being developed under the Air Force program on Space Systems Environmental Interactions Technology. Operation of the system is based on the empirical results of the Space Test Program SCATHA (P78-2) and NASA ATS-6 satellites, both of which successfully demonstrated the principle of safely reducing spacecraft charging levels by the emission of a low energy neutral plasma--effectively shorting the spacecraft and charged dielectric surfaces to the ambient space plasma. The Charge Control System now being designed and built at Hughes Research Laboratories in Malibu, CA., will utilize a xenon-based plasma source capable of igniting within one second, and capable of emitting a quasi-neutral plasma containing more than 1 mA of ions. The spacecraft charging level will be detected by sensors similar to those that operated aboard SCATHA. Satellite frame potential (relative to the ambient space plasma) will be determined by an electrostatic analyzer capable of detecting both ions and electrons in the energy range 50eV-20 keV. Differential charging (relative to spacecraft frame ground) will be determined by a surface potential monitor utilizing two different dielectric surfaces, and covering a range of ± 20 kV with a response time of one second. A transient pulse monitor will detect the presence of exterior spacecraft arcing by measuring its near-field electromagnetic radiation. Automatic operation of the system will be accomplished by a microprocessor controller which will interpret the sensor data and activate the plasma source when predetermined threshold levels are exceeded. With a gas supply for more than 2000 hours of operation in space, the system may be expected to provide on-orbit spacecraft protection for up to 10 years. The system will be completed by the end of 1987, and is expected to be flight-tested at geosynchronous orbit in the 1988-1992 time period.

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INTRODUCTION: It is not necessary to go back much more than a dozen years to get to the earliest identification of the spacecraft charging problem--to 1972 when DeForest [1] published his experimental results from ATS-5 at geosynchronous orbit where he was flying a charged particle detector. During eclipse of the spacecraft by the earth he noticed abrupt dropouts in both low energy electrons and protons, which led him to conclude that the spacecraft was charging to several thousands of volts. Since then a number of spacecraft anomalies have been observed at geosynchronous orbit, predominantly in the midnight to dawn local time sector, and many of these have been attributed to spacecraft charging. A natural mechanism for charging satellites at geosynchronous orbit exists during magnetic substorm conditions. Then the plasma sheet, which contains energetic electrons, is swept inward from the earth's magnetotail, enveloping the spacecraft. Bombarded by energetic electrons, the satellite charges up until a steady-state condition is achieved where the currents to and from the spacecraft are balanced. At that time the voltage level of the vehicle may be comparable to the energy of the impinging electrons; levels of 8 keV can be reached quite readily in this way. (The highest observed value occurred during eclipse in Oct, 1975, when ATS-6 was shown to charge to - 19,000 volts). Dielectric surfaces on the spacecraft can charge at different rates than the vehicle frame ground so that substantial differential charging can be achieved. When this results in arcing on the exterior surface of the spacecraft, optical surfaces can be damaged, thermal blankets can be eroded, and further, the arcs can couple into the command and data lines of the satellite causing false signals, triggering erratic commands, and even destroying solid-state electronics.

By 1979 the U.S. Air Force launched the SCATHA or Spacecraft Charging at High Altitude satellite to near geosynchronous orbit specifically to investigate this problem. The spacecraft was equipped with a full range of sensors to determine the spacecraft state-of-charge as well as the background conditions in space when the charging occurred [2]. Also on board were the Air Force Geophysics Laboratory (AFGL) active experiments - the electron gun and the satellite positive ion beam system (SPIBS) - which could be used to swing the potential of the satellite either positive or negative on command. One of the alternate operating modes of the SPIBS ion gun was simply as a neutral plasma source, and that proved to be the safest and most effective method of reducing both absolute and differential charging on the satellite. These SCATHA results [3], as well as those from the NASA ATS-6 satellite [4] using a cesium plasma bridge neutralizer, have shown that a charged spacecraft, and the dielectric surfaces on it, could be safely discharged by emitting a very low energy (<50 eV) neutral plasma -- in effect "shorting" the spacecraft to the ambient plasma before dangerous charging levels could be reached. This technique forms the basis for our Flight Model Discharge System (FMDS), which is an active charge control system for satellites that will operate automatically in space.

SYSTEM DESCRIPTION: A block diagram of the system is shown in Fig 1. The sensors to detect the spacecraft charging level are basically similar to those that flew on SCATHA, and are comprised of a proton/electron electrostatic analyzer, a surface potential monitor, and a transient pulse monitor. Their outputs are fed to a microprocessor controller, which contains the algorithms to interpret the sensor data, and which determines when the pre-set threshold charging levels have been reached. It then activates the low energy plasma source to discharge the spacecraft. The plasma source is the "heart" of the system -- it will produce up to 1 mA of neutral xenon plasma with energy less than 50 eV, and it will do so within 1 second of being commanded to by the

controller. The controller is the "brains" of the system that coordinates its autonomous operation. It's an 8-bit microprocessor, the 80C85RH, which is a radiation-hardened version for space applications. The electrostatic analyzer detects incoming protons and electrons in 16 adjacent energy channels from 50 eV to 20 keV to provide a measure of the absolute charging of the spacecraft and also to characterize ambient conditions in space that may be conducive to spacecraft charging. Since a negatively charged satellite will accelerate ambient protons up to its potential, the lowest energy channel at which sizeable fluxes of protons are detected would correspond to the absolute charging level of the spacecraft (i.e. relative to the ambient plasma in space). The electron spectrum can provide a predictive capability when an empirical criterion due to Olsen [5] is applied -- namely, when the flux of electrons of energy greater than 10 keV exceeds 10 pA/cm^2, spacecraft charging is likely to occur. The surface potential monitor measures the differential spacecraft charging (i.e. the potential developed on insulating surfaces relative to spacecraft frame ground). Essentially, the back surface voltage of a charged dielectric sample (say kapton, solar cell cover glass, etc.) is measured with an electric field sensor, and that in turn is translated into a front surface potential by means of a prior vacuum chamber calibration. The transient pulse monitor detects arc discharges occurring on the external surface of the spacecraft by counting the pulses and measuring the maximum amplitude and pulse width of the radiated electromagnetic noise bursts. The combination of these three different sensors will provide a good indication of the state of spacecraft charging, but any one alone (by exceeding a present threshold) will be able, via the controller, to activate the plasma source and reduce the charging.

The configuration of the charge control system, as it is currently conceived, is shown in Fig. 2. The 2-liter tank contains xenon gas at 800 pounds per square inch pressure, equivalent to approximately 100 liters at standard temperature and pressure. The gas is fed to the plasma source through a series of valves and regulators maintaining a xenon flowrate to the source of 0.5 standard cubic centimeters per minute. The gas supply is sufficient for more than 2000 hours of operation. The purpose of having two surface potential monitors is to allow for two different dielectrics in attempting to "match" the predominant spacecraft surfaces. Not shown in the figure is the second of two transient pulse monitor antennas, each an aluminum disk 123 square cm in area, about 12.5 cm in diameter, one mounted in the cover plate facing the outside, and the other fixed on the underside of the base mounting plate looking toward the inside of the spacecraft.

The cycling time for the surface potential monitors and the transient pulse monitor are each one measurement per second. The electrostatic analyzer sweep time can be set by ground command for 1,4,8 or 16 seconds for a full scan across the 20 keV energy spectrum, and is set for the smallest period that provides a satisfactory particle count rate. Even when requiring that a measurement exceed the pre-set charging threshold for three successive times (to minimize false alarms), the system response time from the first instance of exceeding the charging threshold to the emission of neutral plasma to discharge the spacecraft can be as little as 5 seconds.

PLASMA SOURCE: Considering the key components of the system in somewhat more detail, a schematic drawing of the plasma source is shown in Fig 3. This is a hollow cathode device whose major features are the thin-walled tantalum cathode tube, a two-metal rolled foil insert of rhenium and platinum impregrated with a barium carbonate low-work-function emissive compound, a

tungsten cathode tip, a keeper electrode which acts as an accessory anode, two azimuthal rings of samarium cobalt magnets which provide a divergent axial magnetic field (in fact a cusp field between the cathode tip and the anode plate), the anode at the top of the discharge chamber, and beyond that a grounded plate at cathode (common) potential. Common potential is connected to spacecraft ground through current detectors which measure the plasma-source net emission current.

In operation neutral xenon gas is fed into the discharge chamber through the cathode tube. Initially, to guarantee fast ignition, a brief burst of xenon gas at high pressure (several hundred Torr) is admitted, and approximately 1000 volts is applied between the keeper and cathode. The arc heats the foil insert to hollow-cathode ignition temperature (about 900° C) in less than a millisecond, and the discharge transitions from arc to hollow-cathode mode. Electrons are emitted thermionically from the low-work-function-coated foil insert in the cathode tube, and then pass through the cathode orifice into the main discharge chamber. In this region, the configuration of the strong magnetic field limits their direct access to the anode, but they do diffuse there by means of collisions. Many of these collisions are with neutral xenon atoms, resulting in the formation of new electron-ion pairs. Ions formed in the discharge chamber follow electric field lines to the cathode-potential surfaces -- at the upstream end of the discharge chamber they bombard the cathode itself, maintaining the proper operating temperature; at the downstream end many of the xenon ions leave the plasma source through the apertures in the anode and ground screen. Electrons also leave the source through these apertures at whatever rate is required to maintain the overall spacecraft/plasma source charge and current balance. (This balance is achieved automatically by the basic properties of the plasma discharge). Ion currents of up to 1mA can be produced in this manner, and if required they may be throttled back to lesser amounts by controlling the keeper power supply. Normal operation of the plasma source will require about 12 watts of spacecraft power, but it will be on only a few percent of the time.

<u>ELECTROSTATIC ANALYZER</u>: Actual vehicle potential (i.e. frame potential relative to space potential) is determined by the data from an electrostatic analyzer (ESA) which measures the distribution of ion and electron energies which are incident on the spacecraft. The basic design of the ESA assembly is shown in Fig 4. Incident particles must pass through the cylindrical plate analyzer section in order to reach the channeltron (channel electron multiplier) and be counted. By stepping a specific sequence of voltages (up to $\pm$200V) across the plates, the analyzer essentially selects out particular energy bands of particles that will be transmitted in the proper curved trajectory to enter the channeltron aperture. There is one background channel and 15 logarithmically spaced energy channels (for ions and electrons each) to cover the range of 50eV to 20keV. A 50V bias (+ for protons, and - for electrons) is applied to a grid at the exit collimator to prevent entry of particles from the plasma source. There is also a 500V post-acceleration bias (this time - for protons, and + for electrons) applied to the channeltron funnel to increase detector efficiency for low energy particles. The energy resolution of each channel ($\Delta E/Eo$) is approximately 39%, and the geometric factor is 6×10^{-4} cm^2-sr. Sweep times of 1,4,8, and 16 seconds are possible, selected by ground command, to insure adequate count statistics in each energy bin.

Interpretation of the ion spectrum in terms of the level of spacecraft

charging is accomplished in the controller using the "distribution function algorithm" (DFA) devised by Spiegel and Cohen [6]. Essentially, this algorithm examines adjacent ESA energy channels to find an abrupt increase in ion counts from one channel to the next. The algorithm calculates the difference of counts between adjacent channels and determines whether this difference is both large enough and of adequate statistical significance to indicate a shift in vehicle potential. The characteristic sharp edge in the ion energy spectrum, below which there would ideally be few ion counts, is taken to be the potential between the spacecraft frame and local space plasma. The DFA has been "tested" against a data base of approximately 10,000 individual ESA spectra from the SC9 experiment aboard SCATHA, and for a charging level of -500V performed to better than 98% compared with spacecraft potential estimates obtained by visually reading the spectrograms.

<u>SURFACE POTENTIAL MONITOR</u>: In order to detect the differential charging of dielectric surfaces on the spacecraft, the system includes a surface potential monitor (SPM) -- actually two of them in order to have two different dielectrics with which to "match" the characteristic exterior surface dielectrics on the vehicle. One of the main factors in the design of the SPM is the requirement of not altering the charge buildup of ions or electrons on the dielectric material due to the measurement itself. Because of this an electric field sensing device is typically used. Our SPM is based on the NASA Lewis surface voltage sensor designed by Sturman [7], and is shown in Fig. 5. The collector plate with the charged dielectric material on its surface is insulated from the rest of the instrument by a ring of Delrin. No electrical connection is made to this electrode. Sensing of the field is done by a vibrating electrode driven by a tuning fork. Directly above the sensing electrode is an aperture plate containing a hole through which it can "see" the electrostatic field created by the charge on the collector plate. As it vibrates, the sensing electrode generates a displacement current that is proportional to the net field, and at the vibration frequency. The phase of this signal is determined by the polarity of the net field. The field at the sensing electrode is nulled to zero by driving the aperture plate to a voltage inversely proportional to that producing the field. By proper selection of the geometry, particularly the hole size in the aperture plate, this attenuation constant can be selected to allow for nulling the maximum field with a maximum of 10 volts applied to the aperture plate electrode. Several modifications to the basic NASA design involve adding a second compensating electrode with a concentric aperture, but of different size. This allows for automatic range switching to full-scale ranges of 0 to ± 2kV and 0 to ± 20kV. Also, an additional grounded shield electrode is installed between the compensating electrode (aperture plate) and the sensing electrode. This prevents any spurious coupling of the electric field emerging from the compensating electrode into the sensing electrode leads. It also makes the instrument's calibration less critically dependent on the centering of the sensing electrode relative to the apertures in the compensating electrodes.

<u>TRANSIENT PULSE MONITOR</u>: The essential problem in detecting the arcs that may result from spacecraft charging is in discriminating between their signals and those due to spacecraft circuit transients. At the moment we know of no unique characteristics in the arc signal that would allow for a single sensor to discriminate between an arc and spacecraft-generated noise. However, if we use two sensors, one looking external to the spacecraft and one internal, and we assume that the spacecraft will act as an imperfect Faraday shield between the two sensors, then for a spacecraft surface arc we can expect that the signal amplitude received at the external sensor will be significantly greater

than that received at the internal sensor. Using the electrostatic field of the arc, which can be detected by capacitive coupling to an electrometer plate, experimental results at JAYCOR (M. Treadaway, private communication, 1984) have shown that the external to internal amplitude ratio can be between 10 and 100 to 1. This is the basis for our using two signal sensors, internal and external, which both detect the electrostatic field of the arc. The sensor is shown in Fig. 6, and consists of a 123 cm^2 plate driving a buffer amplifier with a bandwidth of about 250 Hz through 70 MHz. The sensor/buffer acts as a capacitive potential divider from the electrostatic field of the arc to the spacecraft frame. Two scaling input capacitors to ground are selected by a relay to provide two dynamic ranges of 10 to 300 V/m and 300 to 10,000 V/m. The buffer drives analog circuits which detect positive and negative pulses above a commandable threshold setpoint, the pulse width of signals above threshold, and the positive and negative peak amplitudes. Pulse counts are limited to once per millisecond to protect against ringing. There are two dynamic pulse width ranges, one scaled for 0 to 0.3 μsec and the other for 0 to 10 μsec. The analog signals are converted to digital data and stored in memory under control of a small dedicated microprocessor, which also contains algorithms for interpretation of the data.

CONTROLLER: The controller design employs two separate 80C85RH microprocessors: one services the ESA, producing a "vehicle potential" signal; the second, the "master" microprocessor, serves functions of operating the plasma source, maintaining command and telemetry contact with the spacecraft, and determining when to activate the plasma source (based on inputs from the TPM, ESA processor, and SPMs). This two-processor approach is required because of the heavy computational load imposed by processing ESA spectra to determine the vehicle potential. Electrically alterable read-only memory (EAROM) is used instead of the combination of ROM and RAM for containing default set points, logic decision thresholds, and the oprating algorithms themselves. Advantages of this approach are that the amount of RAM required is minimized; the probability of bitflip - generated hazards to the system is reduced; virtually all parts of the software can be rewritten from the ground, rather than just selected set points; and a system reset or power outage will not cause unwanted default set points to reappear. The software is written in a high-level compiled language (C), which facilitates code maintenance and produces fast and comparatively efficient code. A watchdog timer is included in the controller hardware to guard against loss of processor control which might occur as a result of a processor soft error or a coding error. The watchdog timer is set by an executive routine each time the processor is interrupted (once per second). If the watchdog timer is not reset after 4 seconds, it resets the processor, causing a complete controller reinitialization and software restart.

SUMMARY: A summary of the charge control system physical characteristics is shown in Fig. 7. The system is now being designed and built under contract to the Air Force by Hughes Research Laboratories of Malibu, California. The breadboard model of the system has already been fabricated and successfully demonstrated. The flight units are scheduled for delivery by December, 1987. Prior to that they will undergo qualification tests in a "simulated charging environment" in addition to the normal shock, vibration, and thermal tests. They will be operated in a vacuum chamber at 5 X 10^{-7} Torr; bombarded by energetic electrons in the range 0-30keV, and protons in the range 0-15keV; and illuminated (or eclipsed) from a UV source. However, true qualification of the system can only be obtained aboard an operating spacecraft, preferably at geosynchronous orbit or beyond. We are now looking for just such a test-flight for the 1988 to 1992 time frame.

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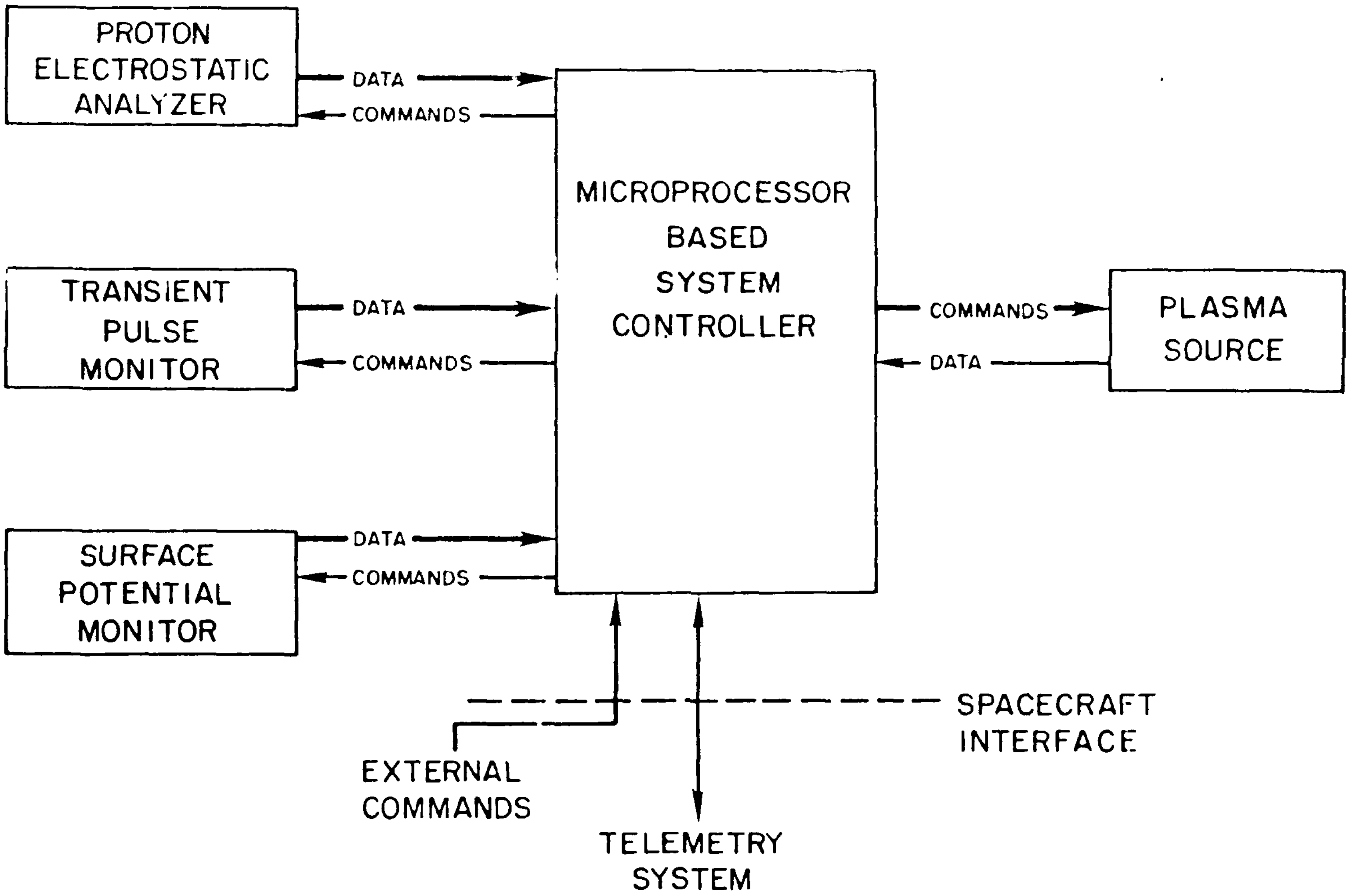
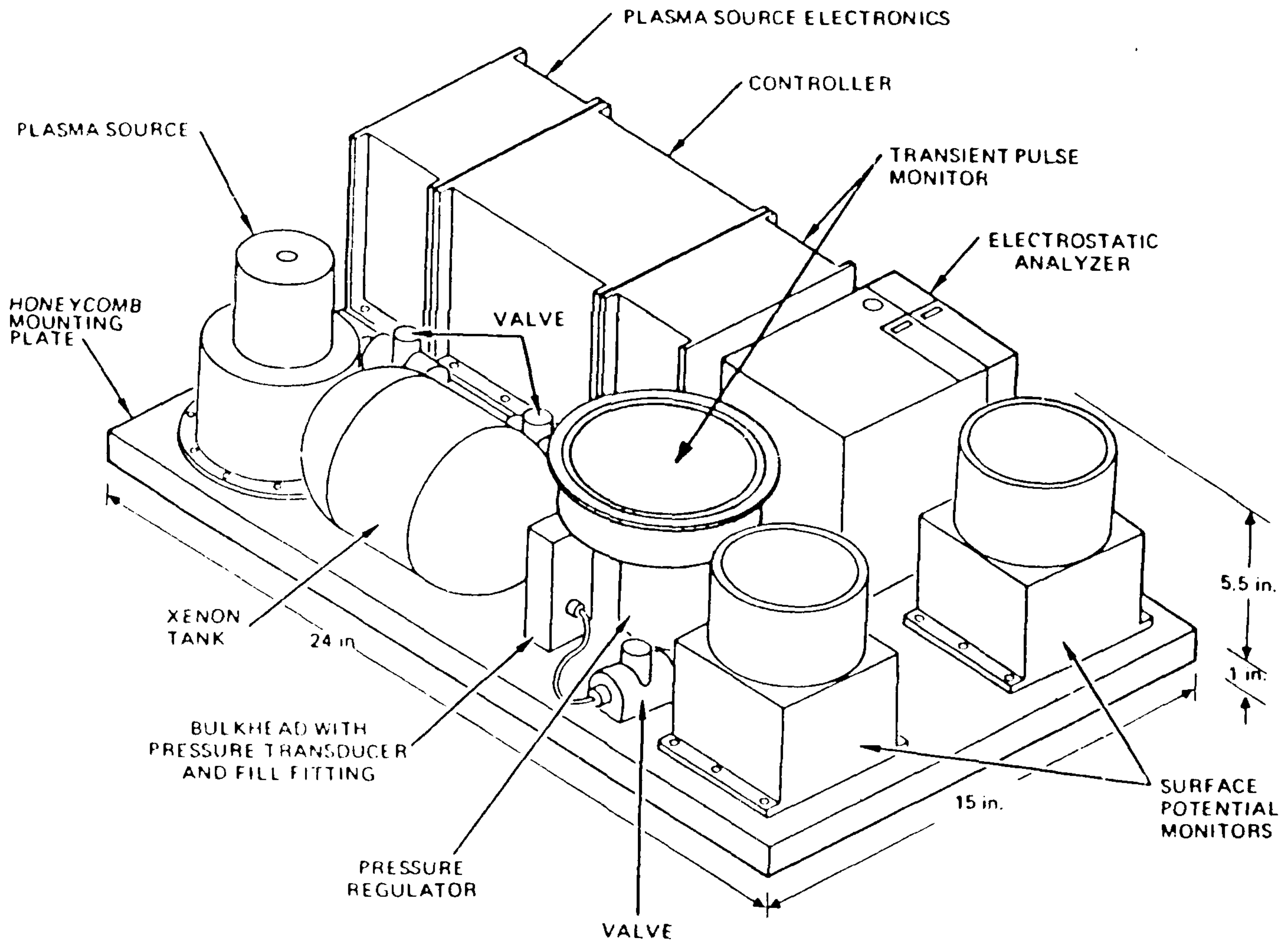
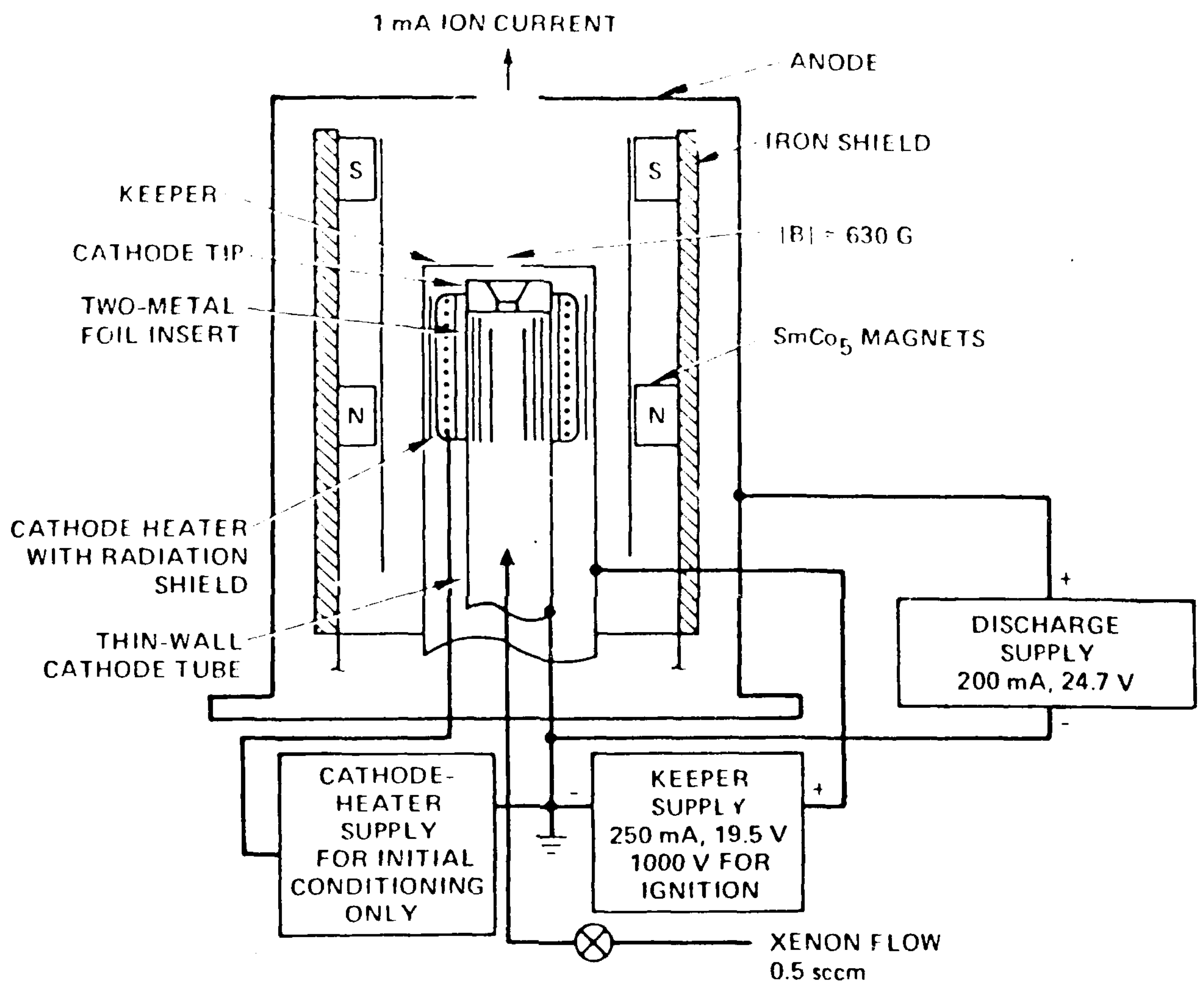
FLIGHT MODEL DISCHARGE SYSTEM

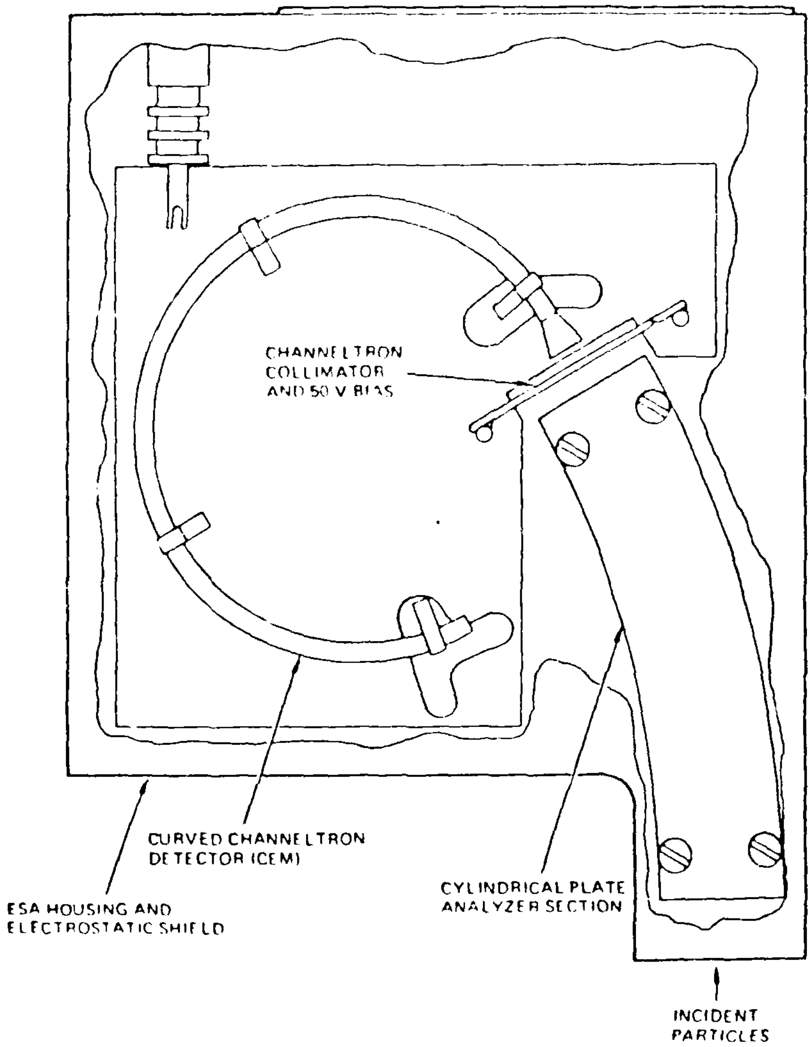


CHARGE CONTROL SYSTEM





PLASMA SOURCE



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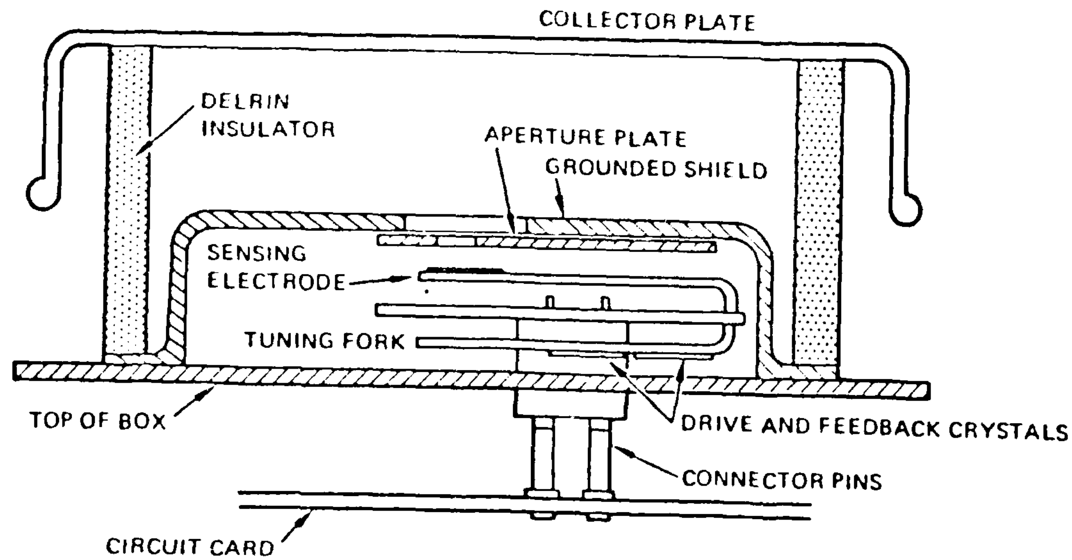
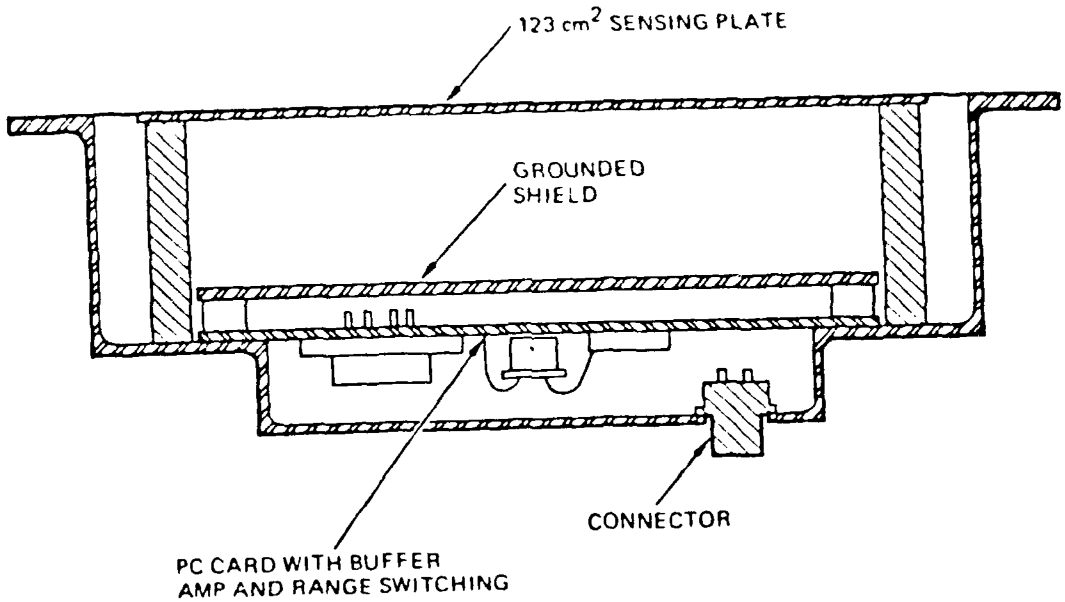


NASA

SPM SENSING HEAD



TPM SENSING HEAD

CHARGE CONTROL SYSTEM SPECIFICATIONS

WEIGHT: 11.5 KG

SIZE: 61 X 38 X 17 CM

POWER: PLASMA SOURCE OFF 13 W
PLASMA SOURCE ON 25 W

TELEMETRY: 143 8-BIT WORDS PER SECOND

COMMANDS: 20 SERIAL-DIGITAL (16 BIT WORDS)

PLASMA SOURCE: UP TO 1mA OF IONS (XENON) OUT
GAS FOR 2000 HOURS OF OPERATION

ELECTROSTATIC ANALYZER: DETECTS BOTH ELECTRONS AND PROTONS
15 ENERGY CHANNELS PLUS BACKGROUND CHANNEL
RANGE 50 EV TO 20KEV
SWEEP PERIOD SELECTABLE 1, 4, 8, 16 SECONDS

SURFACE POTENTIAL MONITOR: 2 DIELECTRIC SURFACES
2 SENSITIVITY SCALES (0 TO ± 2kV, 0 TO ± 20kV)
RESPONSE TIME 1 SEC

TRANSIENT PULSE MONITOR: ELECTRIC FIELD STRENGTHS 10KV/M DOWN TO 10V/M
PULSE WIDTHS 20 nS TO 10 μS
PULSE COUNTS ONCE PER MILLISECOND
COMPARES AMPLITUDE RATIO EXTERNAL/INTERNAL

CONTROLLER: DUAL MICROPROCESSORS (80C85RH)
HIGH-LEVEL LANGUAGE (C)
USES ELECTRICALLY ALTERABLE ROM (EAROM)

Fig 7